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Pluto is the last known planet in our Solar System awaiting spacecraft reconnaissance. In its eccentric orbit taking it 50 AU from the sun, Pluto presently has a thin atmosphere containing methane, which is projected to "collapse" back to the icy planet's surface in about three decades, following Pluto's 1989 perihelion pass at 30 AU. Based on ground and Earth-orbit-based observing capabilities limited by Pluto's small size and extreme distance, present top-priority scientific questions for the first mission concern Pluto and Charon's surface geology, morphology and composition, and Pluto's neutral atmosphere composition and structure.

Budgetary realities preclude a large, many-instrument flyby spacecraft, while distance and launch energy requirements preclude any but the smallest orbiter using presently available launch vehicles and propulsion techniques. A NASA-sponsored Pluto Mission Development activity began this year. Two alternative cost-constrained mission implementations are described, based on which a primary implementation will be chosen. The Pluto Fast Flyby (PFF) mission utilizes an 83 kg (dry) spacecraft launched in 1998 aboard a Titan IV(SRMU)/ Centaur for an ~7 year direct trajectory to Pluto. Instruments described are an integrated CCD-imaging/ ultraviolet spectrometer, with a possible integrated infrared spectrometer. The larger Pluto-350 spacecraft, ~316 kg (dry), carries a broader instrument set, greater redundancy, and requires >11 year flight time launching in 2001 aboard a Delta or Atlas, toward Earth and Jupiter swingbys to provide the energy to reach Pluto. Launch by Proton is under consideration.

Both mission implementations store data during the brief encounter, to be played back over several months. Cost is the primary design driver of both alternatives, with major tradeoffs between spacecraft development, launch services, radioisotope thermoelectric generator procurement and launch approval, and mission operations. Significant benefits are apparent from incorporating "microspacecraft" technologies from Earth orbiters.

Introduction

The objective of the present Pluto Mission Development activity is to be satisfied by a scientific understanding of Pluto and its environs at a level comparable to that of the other outer planets, enabled by data returned from reconnaissance spacecraft. The present implementation options are driven by 1) the need for limited expenditures to meet today's fiscal constraints, 2) the need to meet key scientific measurement objectives to make best use of our first opportunity to visit Pluto and Charon with spacecraft, and 3) the need for a high probability of success in obtaining the highest priority measurements.

Work to date, which began in 1992 January, has focused on two end-to-end flyby mission concepts, with initial examination of an orbiter mission. Developmental design has been undertaken with cost foremost in mind, followed by schedule and performance as secondary criteria.

Two alternative spacecraft have been defined for different mission implementations. Launch vehicle selection, mission operations and science instrumentation are likely to be rather different depending on whether a 100 kg-class spacecraft is chosen instead of a 300 kg-class. The smaller, dubbed the Pluto Very Small (PVS) spacecraft, flies what we have termed the Pluto Fast Flyby (PFF) trajectory from Earth to Pluto off a Titan-IV/Centaur with two solid propellant upper stages. If the Solid Rocket Motor Upgrade (SRMU) is available, flight time is ~7 years; otherwise flight time is ~8 years. The larger, referred to as Pluto Flyby 350 (PF350), carries a broader instrument set and greater redundancy. In a tradeoff of launch vehicle cost on one hand vs. flight time, spacecraft size and cost on the other hand, the Pluto-350 baseline uses Earth and Jupiter swingbys to achieve a flyby 11+ years after launch, but cannot best be launched until 2001, aboard a Delta or Atlas.

A Pluto Orbiter mission was examined with a 35 kg baseline spacecraft and 16+ year flight time. It was to follow a similar Earth-to-Jupiter path as the PF350, with a liquid propellant stage to execute a deep space maneuver between launch and Earth flyby. The same solid rocket motors which are to propel the PFF mission on their way to Pluto were instead to be used to slow the Orbiter down just before reaching Pluto, after which a small onboard propulsion subsystem would insert the spacecraft into orbit. Unfortunately, with reasonably projected technology availability between now and the 2001 launch, it was not possible

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to provide a 35 kg spacecraft with useful science capability or sufficient reliability. For the 100+ kg required to support a useful payload and return its data, flight times exceed 18–20 years, making reliability difficult to achieve and dramatically reducing the mission's perceived value. Work on the Pluto Orbiter has been discontinued in favor of the more viable flyby options.

This paper serves as a status report about halfway into the first year of work toward the first mission to Pluto. Some details and option choices will have changed between preparation in June and presentation in September, as the mission development activity is moving at a fast pace. The actual mission implementation method selected will be somewhat of a blend between the PFF and PF350 options described here. Neither represents precisely the spacecraft which is expected to fly, but comparison of the two serves to illustrate many of the difficult choices which are required to mount an ambitious, and by any accounting, costly, mission.

To provide a sense of size and assist with engineering layout of spacecraft components, heat sources, the antenna and various sensors, a full-size mockup of the present PFF spacecraft baseline has been built by students at the California Institute of Technology and is expected to be on display in Washington at the time of this conference. These and other students are integral members of the Pluto Team, working with engineers and scientists at JPL, University of Colorado, Science Applications International Corp., NASA Headquarters, Southwest Research Institute and elsewhere.

Science Overview

Since Clyde Tombaugh's discovery of Pluto in 1930, very little has been learned about its nature. Pluto's inclined and eccentric orbit of the sun carries it between 30 and 50 AU, so Pluto exhibits a wide seasonal range. Since its orbital period is 248 years, only a short portion of Pluto's year has been sampled. An excellent summary of present-day knowledge and uncertainties about Pluto is contained in the review paper by S. A. Stern [1], from which much of what follows is drawn. Pluto is known to have a thin atmosphere and a relatively large moon, Charon, orbiting at a distance of about 20,000 km. Methane is a constituent of the surface and atmosphere but little else is known about other components. Interest in Pluto has increased since the 1989 *Voyager 2* encounter with Neptune's moon Triton. Triton is a near twin of Pluto in size and albedo and has revealed an extremely complex geology, active surface eruptions, polar ice caps, seasonal volatile changes and limb hazes. Only a spacecraft encounter can provide this kind of information. Pluto is now just past perihelion; as it moves away from the sun its atmosphere is condensing. It is essential that Pluto be explored before the 2020s when its atmosphere will be frozen onto its surface for the next two centuries.

Key questions about Pluto and Charon concern the origin of this "dual-planet" system and its relationship to the rest of the Solar System. While reasonable theoretical limits have been proposed, it is impossible to resolve the mass and density of either body separately—only the aggregate is known. The radii of the two bodies are reasonably well estimated based on the mutual occultation events measured during the 1980s. From the combined system mass we can infer that Pluto has a substantial rocky component. A very tenuous atmosphere containing methane has been detected around Pluto using a stellar occultation, while water ice is indicated on Charon and methane ice on Pluto. At surface temperatures of perhaps 40K, methane ice relaxes over geologic time scales for larger topographic features, while water ice behaves more like terrestrial rock. Thus, there is "the speculative but interesting possibility that Pluto's surface may harbor only the record of more recent impacts, while Charon's harbors a long-term integrated flux. One awaits a spacecraft mission to learn if this indeed is the case" [1].

Ground-based measurements have shown that Pluto's surface reflectance varies, with some longitudinal variations and asymmetrical polar caps [2]. Charon is also thought to have at least subtle surface markings. With at least a transient atmosphere, there is a mechanism on Pluto for material transport, such as by frost sublimation. On both bodies, radiation effects may cause surface chemistry changes resulting in color and brightness variations beyond what would be caused by impacts alone. No doubt there is much to learn, just as with every other first planetary encounter!

According to Alan Stern, "Any viable theory for Pluto's origin must provide a self-consistent explanation for the major attributes of the Pluto-Charon system. These include: 1) the existence of the binary's exceptionally low, ~6:1 planet:satellite mass ratio; 2) the synchronicity of Pluto's rotation period with Charon's orbit period (6.39 days); 3) Pluto's inclined, elliptical, Neptune-resonant orbit; 4) the high axial obliquity of Pluto's spin axis and Charon's apparent alignment with it; 5) Pluto's small mass ($\sim 10^{-4}$ of Uranus and Neptune); 6) Pluto's high rock content—the highest among all the outer planets and their major satellites; and 7) the dichotomous surface compositions of Pluto and Charon" [1]. Using Hubble, Extreme Ultraviolet Explorer, and other satellite and ground-based data, more information about Pluto and Charon will be deduced before the first spacecraft arrives, but any successful mission will

revolutionize our knowledge of the planet and its environs.

The science goals and measurement objectives for a first reconnaissance mission to Pluto were formulated and prioritized by NASA's Outer Planet Science Working Group (OPSWG), as noted in Table 1. The three category "1a" science objectives were identified as the highest priority required for this first mission, with the "1b" and "1c" category objectives considered desirable but non-essential.

Table 1 — Pluto Core Measurement Objectives (no ranking intended within categories).

Neutral Atmosphere	1a
Geology and Morphology	1a
Surface Composition Mapping	1a
Ionosphere	1b
Bolometric Bond Albedo	1b
Surface Temperature Mapping	1b
Energetic Particles	1c
Bulk Parameters (R, M, ρ)	1c
Magnetic Field	1c
Additional Satellites	1c

The goal of an initial reconnaissance of the Pluto-Charon system is to understand the physical and geological processes on the surface, the surface composition, and the composition and nature of the atmosphere. Our present thrust is to test the hypothesis that these top-priority objectives can be met within the cost, size, and mission time constraints of a very small spacecraft mission. To do so will require new developments in instrumentation, described in the "Pluto Very Small Implementation Option" section. With modest coverage of the Category 1a objectives, the PVS option could provide comparable or better scientific coverage of Pluto and Charon than was provided by Voyager at Triton.

Alternatively, a more comprehensive "strawman" payload, also defined by the Outer Planet Science Working Group, is being utilized to drive the PF350 design. With up to five instruments and a 30 kg mass allocation, new instrument development will be required for this implementation, but perhaps less aggressive than required for the PVS option in the case of any single instrument. With more instruments and greater complexity, overall spacecraft and instrument costs will be higher for the PF350 option, and arrival at Pluto would be much later, but science coverage would be much more comprehensive than provided by Voyager at Triton.

Launch Vehicle and Trajectory Analysis

Synopsis

To arrive at Pluto early, only direct trajectories have low flight times and avoid the need to await proper alignment of Jupiter. The PFF baseline takes advantage of these benefits. Such missions require a large launch vehicle (Titan- or Proton-class) and extra stages with a small spacecraft. Once spacecraft mass is too high for an attractive direct flight time, as in the PF350 baseline, then smaller launch vehicles (Delta- and Atlas-class) may be used along with differing Jupiter gravity assist trajectories, with the disadvantages of constrained launch date, late arrival, and long trip time.

Interplanetary Trajectory Options [3]

A wide range of trajectory types to Pluto are available for missions using low-mass spacecraft in the late 1990s through the early 2000s. The goal is to select the trajectory which will minimize flight time, specific launch energy (C_3), and post-launch ΔV while providing desirable launch and back-up opportunities, especially if two spacecraft on separate launches are to be considered. A key consideration is specific launch energy, since that, along with the spacecraft mass, dictates the set of launch vehicles which can be used for a given flight time.

Only ballistic (high-thrust) trajectories were considered in this analysis; these transfers include: direct (DIR), Jupiter Gravity Assist (JGA), two- and three-year Earth Jupiter Gravity Assists (2,3-ΔVEJGAs), and combinations of Venus Earth Jupiter Gravity Assists (VEJGAs). While solar-electric and nuclear-electric (low-thrust) systems are estimated to be able to provide better performance than chemical propulsion, they have not been included in this analysis since schedule and cost uncertainties make these technologies unlikely candidates for these cost classes.

Figure 1 shows launch opportunity vs. C_3 for the various trajectory types discussed below. The shading indicates approximate post-launch ΔV requirements for flight times of approximately 7–15 years.

Direct

Conceptually, the simplest trajectory goes direct from Earth to Pluto. Since no gravity assists are used, there is a yearly launch opportunity. Figure 2 is a plot of C_3 vs. flight time for direct trajectories to Pluto launching in 1998 and 2004. As shown, the difference in launch energy between the two opportunities is small, so back-up trajectories exist which actually have greater mission performance. The problem is the actual value of launch energy. Fast flight times require large launch energies; few launch vehicles can

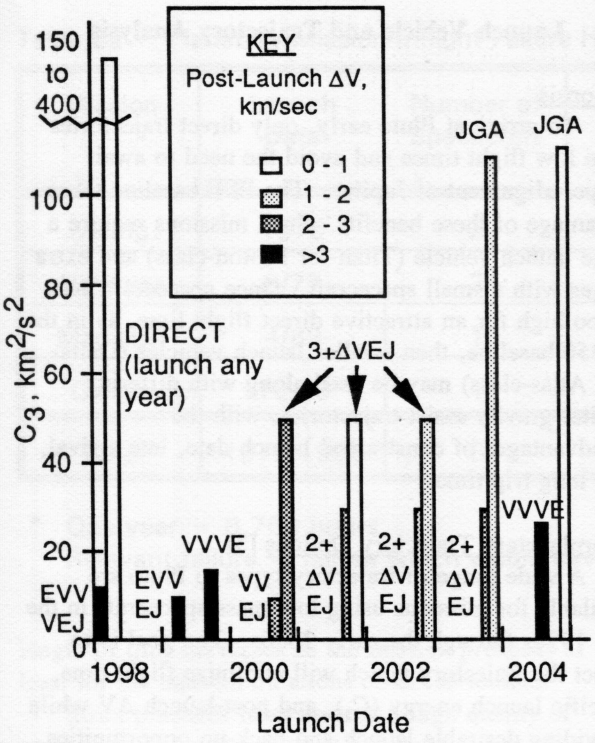


Figure 1 — Pluto Flyby Trajectory Types.

inject mass to C_3s much over 110 km²/s², so upper stages are required [4]. Alternatively, in order to have a fast flight time without augmenting the launch vehicle, gravity assist trajectories must be used.

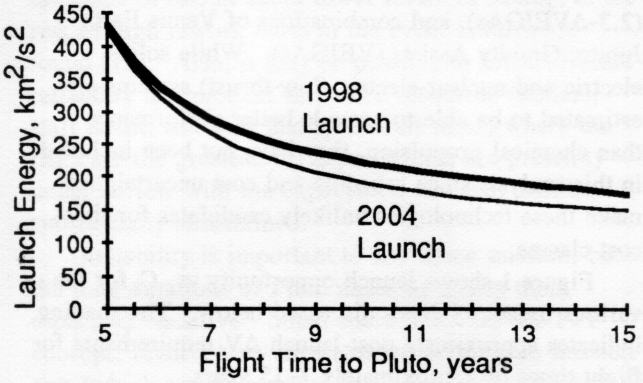


Figure 2 — Launch Energy vs. Flight Time for Direct Trajectories to Pluto.

Direct with Jupiter Gravity Assist

Jupiter is the only outer planet with the proper orbital phasing and mass to provide a beneficial gravity assist to Pluto in the timeframe of interest. By launching to Jupiter enroute to Pluto instead of going to Pluto directly, the specific launch energy is reduced to the 100–120 km²/s² range. Relative motion of Jupiter and Pluto yields only three JGA launch years per Jupiter-Pluto synodic period (roughly 12.5 years); the next set occurs in 2003–2005. As shown in Figure 1, of these opportunities, the 2004 launch is the best since flight time, post-launch ΔV and specific launch energy are minimized for a conservative Jupiter flyby constraint (see below).

Jupiter Flyby Distance Constraint. Jupiter has a severe radiation environment. The potential for accumulating significant electron and proton radiation doses can be quite high inside 14 R_J and could cause lasting damage to the spacecraft and its instruments. There is also danger of single-event upsets due to high energy particles which may ultimately cause no permanent damage to the parts but may cause errant computer commands or data loss. Missions have flown as close as 5 R_J; however, the degree to which the spacecraft were prepared (or "rad-hard") determined to a large extent how well they fared. Spacecraft can be made rad-hard by using radiation shielding or by purchasing parts which can take higher doses.

The constraint on Jupiter flyby distance, therefore, includes three key considerations: shielding mass, cost of rad-hard parts, and flight time. Figure 3 shows how, for a given trajectory and launch opportunity, the flight time to Pluto is determined by the allowed Jupiter flyby radius. There is a ΔV cost for constraining the trajectory to a particular flyby radius and making up the flight time with propulsive burns; however, the severity of this ΔV penalty is such that it precludes significant reductions in flight time.

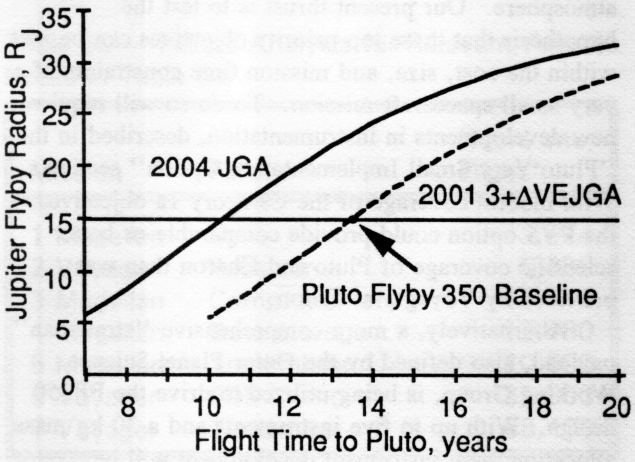


Figure 3 — Flight Time to Pluto vs. Jupiter Flyby Constraint.

Analyses have begun to investigate the Jupiter radiation trade-space better; for now, however, in order to keep a low-mass, low-cost spacecraft, a Jupiter flyby constraint of 15 R_J has been used. If further analysis shows that a 10 or 5 R_J flyby is

feasible within the cost and mass constraints, significant flight time reductions could be realized.

Two- and Three-Year ΔVEJGAs

Specific launch energies for the JGAs are still quite high, requiring the use of more capable and more expensive launch vehicles. It can be beneficial to add an Earth gravity assist to the JGA in order to lower the C_3 at the expense of post-launch ΔV and flight time. The key is to launch on a two- or three-year Earth-return trajectory. At apoapse of the Earth-Earth leg, a ΔV is performed which lowers periapse, re-targeting the Earth flyby in order to effect a gravity assist swingby instead of a resonant Earth flyby. The spacecraft then travels on to Jupiter and finally to Pluto. The acronym ΔVEJGA is derived from ΔV, Earth, Jupiter Gravity Assist. While this strategy adds two or three years to the flight time, the specific launch energies are 25–30 km^2/s^2 with minimum post-launch ΔVs >1 km/s for the two-year ΔVEJGA, and 47-52 km^2/s^2 with typical minimum post-launch ΔVs >500 m/s for the three-year ΔVEJGA. This reduction in specific launch energy, despite the additional propellant mass for the post-launch ΔVs, enables consideration of lower-cost launch vehicles such as the Delta II or the Atlas IIAS.

Another benefit of the ΔVEJGAs is the automatic availability of backup launch opportunities. Since the Earth gravity assist can occur pre- or post-periapse, there are two types of ΔVEJGAs, referred to as "+" or "–" ΔVEJGAs. For instance, a 2+ΔVEJGA will launch approximately 45 days before the 2-ΔVEJGA in order to optimize the total energy for a post-perihelion Earth gravity assist (the 2- uses the pre-perihelion gravity assist). While there are small performance differences between a "+" and a "–" ΔVEJGA, these trajectories offer two launch opportunities per year for as many years as there is a beneficial Jupiter-Pluto phasing.

Venus Gravity Assists

The next major reduction in specific launch energy can be provided by a Venus gravity assist. In general, there are many trajectory types which can utilize one or more Venus swingbys resulting in C_3s as low as 13 km^2/s^2. The best opportunity identified to date in the timeframe of interest is a VVEJ trajectory in 2000.

There are three main drawbacks to the VVEJ. First, there is a deterministic post-launch ΔV requirement of >2 km/s for the lower flight times. Second, these trajectories require perihelia of 0.7 AU or less. Since solar flux scales as the inverse of distance squared, the thermal environment for the spacecraft at Venus is twice as severe as the environment at 1 AU. Third, the Venus-Earth synodic period is roughly 1.5 years, which places Earth far from where it should be after one synodic period. Therefore, launch can only occur every other synodic period, or once every three years, in order to go on to Jupiter. In order to match the Jupiter-Pluto synodic period, this only leaves one of these particular VVEJ opportunities and no backups of this type per decade.

Finally, triple-Venus-Earth swingby opportunities have been identified which do not require a Jupiter swingby. The advantage of this type of trajectory is that more launch opportunities are available than with a Jupiter gravity assist since the Venus-Pluto synodic period is much less than the Jupiter-Pluto synodic period. While the C_3s for these opportunities can be less than 20 km^2/s^2, the VVVEGA does require quite a large post-launch ΔV for flight times less than 15 years, making it a less attractive mission.

Solid Rocket Motor Staging

The use of solid rocket motors (SRMs) as additional upper stages is a method of augmenting the overall launch vehicle performance for a particular specific launch energy. Past missions with high specific launch energies, such as *Voyager, Pioneer 10* and *11*, and *Ulysses*, have used SRM upper stages for injection from the Titan/Centaur, Atlas/Centaur, and the Space Shuttle/IUS, respectively. A Thiokol Star 48B gave the 360 kg Ulysses spacecraft an additional 4200 m/s. Staged SRMs have been used many times and are a necessity when considering trajectories with C_3s much higher than 100 km^2/s^2.

Pluto Flyby Mission Performance

Based on information involving trajectory requirements, launch opportunities, and launch vehicle performance estimates, it is possible to create a mission "trade-space." The following paragraphs explain the assumptions behind the performance figures and the process-of-elimination for choosing the two baseline missions. Major disadvantages of each baseline are cited.

Performance Assumptions

Figure 4 is a plot of net dry spacecraft mass vs. flight time to Pluto for a range of trajectory types and launch vehicles. Only the performance for the best launch year of each trajectory type is considered; Figure 1 can be used to determine the availability and relative performance of backup missions to the primary for each trajectory type.

Net dry spacecraft mass. Net dry spacecraft mass is defined as the mass of the spacecraft without propellant or tankage. This method is used to provide a fair comparison between missions. Since the considered trajectory types are varied, the post-launch ΔV requirements can vary substantially between missions. Propellant requirements can quickly exceed

tank capabilities, especially for low-mass spacecraft. Therefore, plots of dry spacecraft mass alone may not accurately represent the fact that the required tank mass varies disproportionately with increasing propellant requirements, potentially leaving an unacceptable payload mass. As in the case of the SRMs, there are a finite number of tanks available for these missions, so actual tank mass was used instead of scaling equations; therefore, the curves are "bumpy" as one tank is switched for another and the ratio of tank to total dry mass changes.

Other performance assumptions include:

Propulsion. The spacecraft were assumed to have monopropellant systems except where noted. These systems generally allow an engine I_{sp} of 215 s.

Launch Periods. In order to add more contingency for implementations involving SRMs, those launch periods were selected to be 21 days long; otherwise, the launch periods were 11 days.

SRM Staging. Where SRM upper stages were used, the staging selection was optimized for the launch vehicle capability. No more than two stages were used, since the performance benefit from a third stage generally did not justify the additional cost and complexity.

Venus Opportunities. Figure 4 does not include the Venus opportunities described above due to high post-launch ΔV requirements, the difficult spacecraft thermal environment, and the associated flight times when compared to the other missions.

Jupiter Flyby. Finally, performance of trajectories requiring Jupiter flybys are shown as if the spacecraft were not forced to fly outside of 15 R_J; however, markers are placed on the curves in Figures 4 and 5 to show the 5, 10, and 15 R_J limits as a function of flight time.

Selection of Baseline Missions

Dry masses of the two alternative mission concepts are 83 kg and 316 kg; however, the *net* dry spacecraft masses are 78 kg for the PVS and 297 kg for the PF350. Using these net dry masses, it is possible to narrow down the mission possibilities for each spacecraft.

VSS Baseline Selection. The goal of the VSS mission is to provide rapid science return from Pluto and minimize costs. Therefore, multiple gravity assist trajectories are at a disadvantage, while direct or JGA missions using SRM upper stages offer flight times under 10 years. Figure 4 shows five mission opportunities with flight time under 10 years for a 78 kg net dry spacecraft mass. Two of these use the Proton, and will be analyzed once performance and availability of the Proton is better understood. This

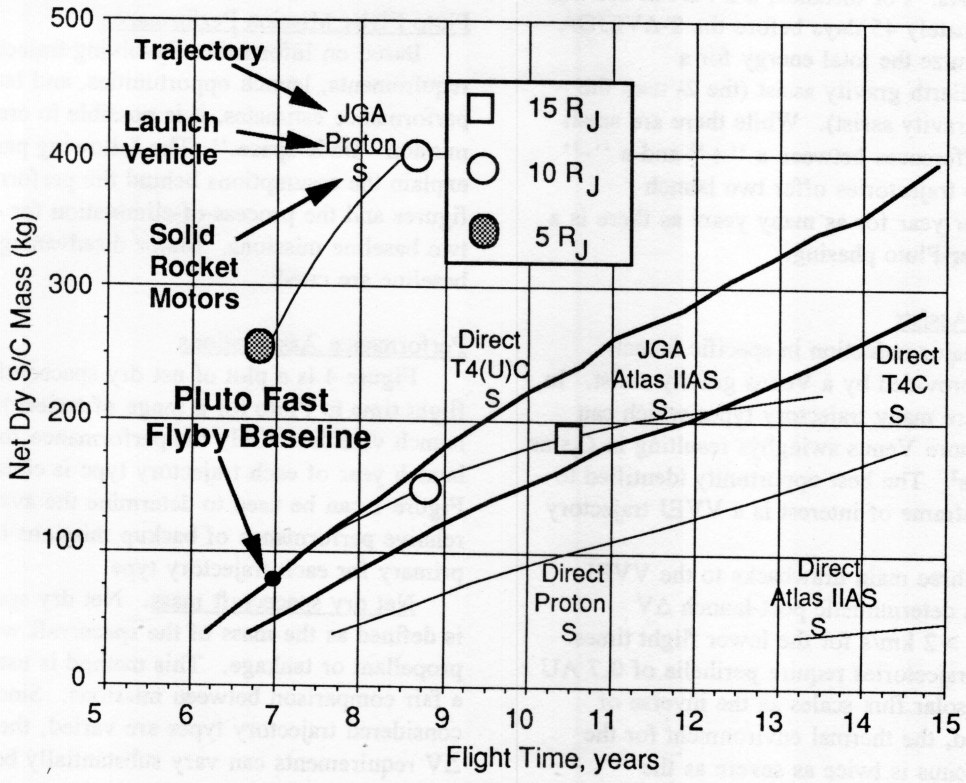


Figure 4 — Pluto Fast Flyby Trade-space: Net Spacecraft Mass vs. Flight Time.

leaves direct trajectories off the Titan IV/Centaur or JGA trajectories off the Atlas IIAS for the following discussion.

While the launch cost of the Atlas IIAS is greatly reduced from that of the Titan IV/Centaur, the flight time to comply with the 15 R_J flyby constraint is about 10.5 years. In addition, launch could not be until 2003–2005. For this reason, the baseline mission is a direct transfer on the Titan IV(SRMU)/Centaur/Star 48B/Star 27, with the Titan IV(SRM)/Centaur as a fallback if the SRMU program does not continue (with a one year penalty in flight time). This implementation allows a seven-year flight time to Pluto and a launch opportunity any year.

The main disadvantages of this baseline mission are the launch vehicle cost and the reduced science return. While launch vehicle cost has typically not been included in NASA's cost classifications, it is still part of the overall mission cost, and therefore should be minimized to the extent possible. Unfortunately, with conventional launch vehicle performance, rapid science return from 30 AU must be traded off against the breadth and depth of science returned. The great reduction in flight time, as well as the opportunity to launch any year, are two of the prime arguments in favoring this implementation.

PF350 Baseline Selection. The PF350 design philosophy entails use of a relatively inexpensive launch vehicle offering more payload mass, thus accepting longer flight time to Pluto. The PF350 spacecraft is fully redundant in a manner similar to *Voyager* (''block redundant'') and is likely to cost much more than the PVS spacecraft. Figure 5 is the PF350 equivalent of Figure 4, with the addition of 2+ and 3+ΔVEJGA trajectories.

Six potential mission possibilities exist. One of these uses the Proton, and may be considered in future work. Two opportunities use the Titan IV/Centaur and are eliminated as a result of the high launch vehicle cost. (In this case, the combination of higher spacecraft cost, high launch vehicle cost, and longer, more expensive operations was considered prohibitive. A PF350 spacecraft on a direct trajectory using the same launch vehicle as the VSS could have a flight time to Pluto of less than 12.5 years.) Three mission opportunities remain; the 2+ΔVEJGA on an Atlas IIAS can be eliminated since flight times less than 15 years violate the 15 R_J constraint. Therefore, the two concepts left are 3+ΔVEJGA trajectories using either a Delta II (7925) or an Atlas IIAS launch vehicle. As shown in Figure 5, the 15 R_J limit for both cases occurs at a 13.5-year flight time. The choice is between launch vehicles, and since the Delta II costs roughly half of the Atlas IIAS, the PF350 baseline mission is the 3+ΔVEJG off the Delta II (7925).

The main disadvantage of this baseline is the long flight time. Spacecraft reliability issues will be a major concern of this mission. Further analysis may determine that the spacecraft can go closer to Jupiter, thus reducing the flight time substantially.

While the 3+ΔVEJGA has a backup 3-ΔVEJGA about 45 days later, another disadvantage of this mission is the fact that the launch opportunities in the following years (2002 3+ΔVEJGA, 2003–2005 JGA) all have different post-launch ΔV requirements and, in the case of the JGAs, drastically different launch energies. It is likely that the Delta II (7925) could not accommodate these later year backups, and the spacecraft would have to be modified to at least carry the propellant loading for the worst year or accept a longer flight time on the backup missions. While this scenario is common to other outer planet missions (e.g., *Cassini*), it can be avoided with a direct trajectory whose characteristics are fairly constant from year to year.

Pluto Very Small Implementation Option

Overview

Numerous ''very small'' spacecraft have carried out successful Earth orbital missions. *Sputnik 1*, at 83 kg, happens to have been the same dry mass as our present Pluto Very Small baseline spacecraft concept. The first space missions were small by necessity—that was all we could launch. A recent resurgence of ''microsats,'' ''lightsats,'' and small spacecraft by other names has been brought about by advances in electronics technology, development of small launchers, wider availability of ''piggyback'' slots for small secondary payloads, and a variety of other reasons. Many of these very small spacecraft have also been produced on very small budgets, such as those designed, built and operated by members of the amateur radio community (e.g., AMSAT's Microsats, Webersat, and the University of Surrey's UOSAT series) and used to test new technologies in communications and instrumentation.

Spacecraft on the order of 100 kg and smaller have been flown on deep space missions (e.g., *Pioneers 6–9* and *Hiten*), and been proposed for asteroid and other missions [5,6]. A series of ~14 kg RTG-powered probes was proposed to search gravitationally for evidence of a tenth planet [7]. Pluto's extreme distance makes time of flight one of the primary impediments to a mission there. However, with *Pioneer 10* and *11* still operating after about 20 years, and the *Voyagers* going strong after 15, it is clear that the outer Solar System is a relatively benign environment for well-built craft. One benefit is that thermal stresses and cycling experienced in the inner Solar System and with

planetary orbiters are largely absent. Whatever our experience, long missions do introduce reliability concerns, and with the primary objective not reached until the end of the mission, long duration can kill the political and financial support necessary to fly.

After many failed proposals for Pluto missions taking 12, 15 or more years, the prospect of a flyby in under 10 years without new propulsion technology was very attractive [8]. The PFF mission implementation concept was developed to determine just how small a spacecraft could be used for a scientifically attractive initial reconnaissance of Pluto while maintaining reasonable expected reliability. The initial 35 kg target was soon exceeded based on the demands of imaging, telecommunications over a reasonable data link, and all the consequent demands on other subsystems. However, the "penalty" of increasing flight time from ~5 to ~7 years was considered a worthwhile price to pay for an attractive mission with an ~85 kg dry mass spacecraft. Another ground rule which drove mass higher was use of components which are reasonably likely to be flight qualified by 1994, and which do not, with some exceptions, require large developmental expenditures charged to Pluto Mission Development.

Pluto Fast Flyby Mission

Figure 6 shows the PFF direct trajectory departing in 1998 February with a seven-year flight time to Pluto. Maximum specific launch energy over the 21-day launch period is 305 km^2/s^2, equivalent to a total ΔV out of low Earth orbit of 12.9 km/s. After the Centaur burn, which delivers about 5.9 km/s, the SRMs are spun up and fired.

Pluto arrival will be designed carefully according to science desires and spacecraft capabilities (see below). The arrival V_∞ (or the relative approach velocity of the spacecraft with respect to Pluto) is 18.2 km/s. The V_∞ of Voyager at Triton was similar; however, the actual Triton flyby speed was 24 km/s due to the influence of Neptune's mass. The low mass of the Pluto/Charon system will cause only a minute flyby velocity increase.

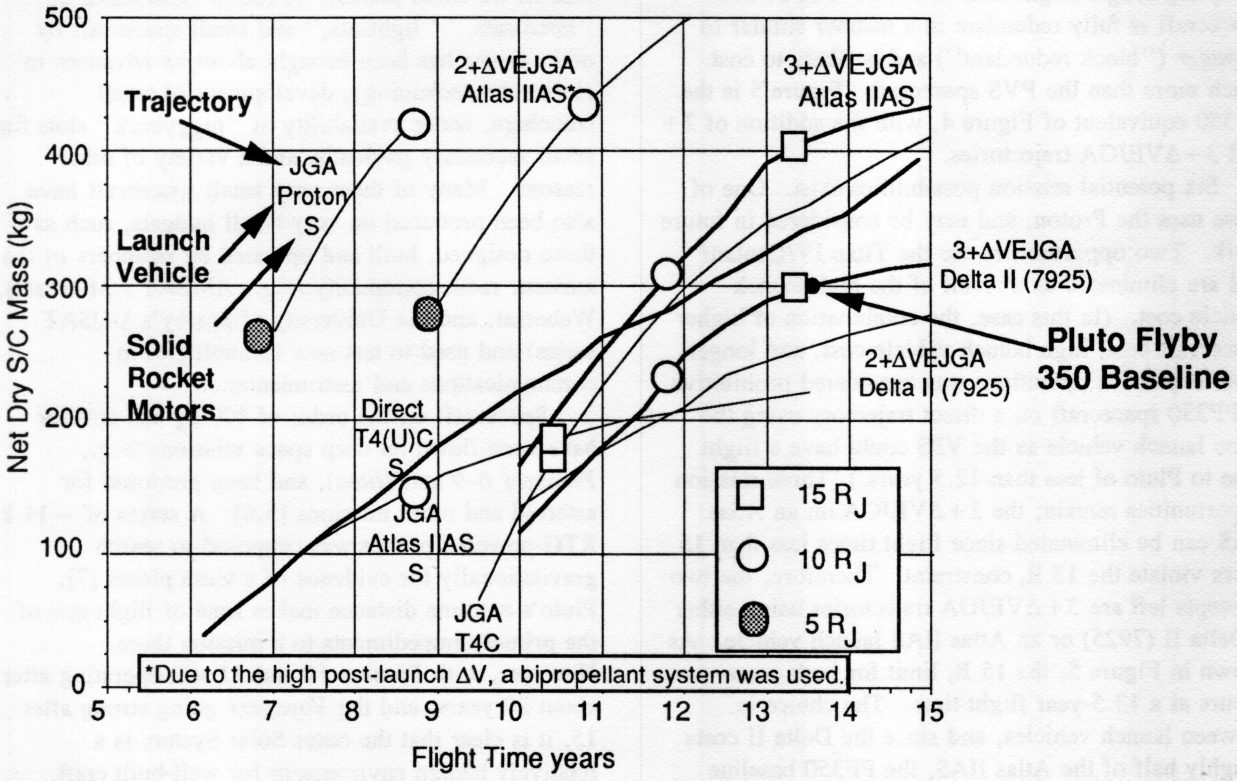


Figure 5 — Pluto Flyby 350 Trade-space: Net Spacecraft Mass vs. Flight Time.

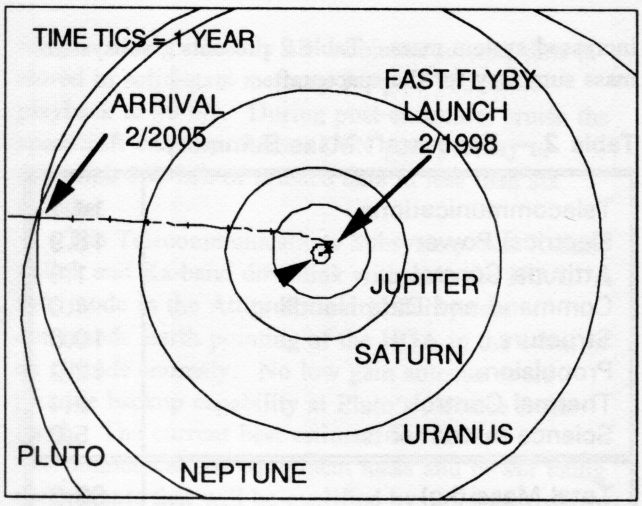


Figure 6 — Pluto Fast Flyby Trajectory.

Science and Instrumentation

In order to achieve the top priority science goals (Category "1a" in Table 1) within the constraints of low mass, power and cost, a fully integrated science payload must be developed. The science payload mass allocation for the PVS spacecraft is in the range of 5 to 10 kg with a power allocation of about 5 W. The drivers behind these low mass and power allocations are that they enable developing missions which fit the Discovery mission spacecraft cost profile and allow fast flight times to Pluto. We are examining the possibility of combining the functions of three conventional instruments, a CCD Camera, Ultraviolet Spectrometer (UVS), and Infrared Spectrometer (IRS), into one fully integrated payload. This payload will share optics, mechanisms, electronics and packaging where possible and has the objective of meeting the Category 1a science goals. Some other goals, such as a search for additional satellites, and resolving separately the mass and density of Pluto and Charon, can be achieved with the same instrument set by adding to the observation sequence.

Initial work has begun on the Camera optical design. Its main measurement goal is to obtain better than 1 km resolution imaging of the illuminated surfaces of Pluto and Charon to study their geology and morphology. The defining parameters are the required resolution and pixel size. By setting the diffraction limited resolution for 600 nm at 10 μrad to fit on a 7.5 μm pixel, the camera design can be derived. A lightweight 750 mm focal length Ritchey-Chretien design is proposed with a 73 mm aperture. By using a 1000×1000 charge coupled device (CCD), the camera would have a field of view of 10 mrad or about half a degree. The resolution of 10 μrad was selected to match the pointing stability of the spacecraft. During the close flyby it is expected that target motion will cause significant smear for exposures longer than about 0.1 s. This smear will be compensated by either target tracking with the spacecraft or by electronically shifting pixels in the CCD. A filter wheel with a minimum of six positions will also be included. Imaging data will be digitized to 8 bits, losslessly compressed (to a ratio of about 3:1) and stored in the solid state memory. Storage for about 100 images is required. The camera will have a read-out time of about 2 s, allowing fast framing during the rapid encounter. This requirement is important because the entire time the spacecraft is in range to achieve <1 km resolution on Pluto is only on the order of 1 hour.

The UVS is included to measure the solar occultation for atmospheric data. Several concepts are being considered. These include a stand-alone UVS with separate optics, sharing the CCD Camera telescope, and using a pickoff mirror adjacent to the CCD focal plane to send a UV signal into a separate UVS focal plane, and others. Final choice of a design will be made on the basis of meeting the Pluto science goals.

Additional measurement capability in the infrared region is required in order to understand surface composition and global distribution of volatiles. This can be done by spectrophotometric mapping of Pluto in the 1–2.5 μm wavelength region. The all-reflecting CCD Camera telescope can be used over a wide range of wavelengths. We are examining the possibility of also incorporating an Infrared Spectrometer into the camera focal plane which could be used simultaneously or sequentially with visible imaging.

Clearly the mass and power constraints of the PFF mission are severe, but the science benefits of arriving at Pluto early enough to measure the thin atmosphere are high. If we are to surmount the technical challenges of developing integrated payloads to meet these specific science objectives, we must focus recent technology thrusts in integrated instrument designs and microdevices into a coordinated effort to directly respond to current spacecraft needs. This approach more closely corresponds to successful efforts made in the early years of NASA to invent technology around specific needs, but also carries its own set of risks.

Conceptual Flight System

The flight system consists of the spacecraft, the solid rocket motor (SRM) stages, and all structural adapters above the separation plane of the Centaur. The SRM stages consist of a Thiokol Star 48B and a Star 27. Included with the Star 48B is a chemical rocket spin-up system similar to that used on PAM-S. The Star 27 has attached to it a nutation control system similar to that used on PAM-D, yo-yos for despinning the empty Star 27 case and spacecraft, and a sequencer to initiate SRM ignitions, separations,

spin/despin, etc. Figure 7 shows the flight system in the Titan IV shroud.

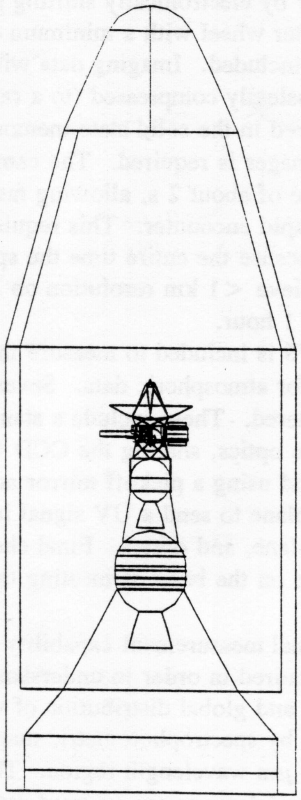


Figure 7 — PVS Flight System Launch Configuration.

The spacecraft was initially described as a largely single-string system to form a simple basis for comparison of design options. Activity is now focused to determine the most cost- and mass-effective implementation of fault tolerance. This effort includes consideration of both block redundant subsystems on a single spacecraft to create a dual string spacecraft and redundancy through two single-string spacecraft on one or two launch vehicles. Component-level redundancy within "boxes" is being considered in some areas, such as solid state memory, where partial capability is acceptable instead of full dual string redundancy. This description focusses on the original single-string baseline, but the reader must recognize that this configuration would never be proposed to fly as-is because of poor reliability.

Figure 8 shows the PVS spacecraft. The high gain antenna (HGA) shown is about 1.5 m in diameter. Overall spacecraft dimensions are ~1.6 m maximum width and ~1.3 m height. The bus has a 0.5 m maximum diameter. Dry spacecraft mass is 83.5 kg including 14.6 kg contingency for mass growth during detailed design. The spacecraft is loaded with 16.5 kg of hydrazine, resulting in a total wet spacecraft mass of 100 kg. Additional mass margin exists in the form of increased flight time with increased system mass. Table 2 presents a subsystem mass summary for the spacecraft.

Table 2 — Spacecraft Mass Summary.

Telecommunications	14.7
Electrical Power	18.9
Attitude Control	1.6
Command and Data Handling	4.0
Structure	10.0
Propulsion	12.2
Thermal Control	2.4
Science Instruments	5.0
Total Mass (kg)	68.8
Contingency (21.4%)	14.7
Monopropellant (400 m/s)	16.5
Total Wet Spacecraft (kg)	100.0

Many features of this conceptual design contribute to lower cost. The overall spacecraft concept has been kept simple. There are no articulations or deployments. A cold gas thruster attitude control scheme was chosen over the complexity of reaction wheels, stable instrument platforms, or a controllable secondary mirror in the CCD Camera. The spacecraft also uses components that will be qualified by 1994, with some exceptions. The radioisotope thermoelectric generator (RTG) is derived from a standard general purpose heat source (GPHS) device as used in the *Galileo* and *Ulysses* RTGs. The propulsion subsystem is entirely off-the-shelf, and the HGA could be modified residual *Viking* hardware. Also contributing to lower cost is the minimal performance of the spacecraft. The low data rate, modest data storage, small ΔV, and minimal power level are all the result of an effort to keep the cost down and capability to the minimum required to accomplish mission objectives.

The flight system is designed to execute the following mission scenario. The Centaur spins the flight system up to ~10 rpm prior to separation. Additional spin-up to ~60 rpm, SRM burns and separations, nutation control, and yo-yo spin-down are sequenced by the propulsion "stack" sequencer on the Star 27. After release from the Star 27, the spacecraft acquires an inertial star reference, turns the HGA to Earth point and establishes communications. After performing an injection error correction maneuver, the spacecraft cruises with the HGA Earth-pointed and uses one 8-hour Deep Space Network (DSN) pass per week. At distant encounter, optical navigation images are taken by the Camera and returned to the ground

for processing. Nearly all encounter science data is stored in solid-state memory for post-encounter playback at 40 b/s. During post-encounter cruise the spacecraft uses one 8-hour DSN pass per day to downlink 200 Mb of science data in less than six months.

The Telecommunications Subsystem uses X-band uplink and Ka-band downlink with only the HGA. A safe mode in the Attitude Control Subsystem commands Earth pointing of the HGA in the event of an attitude anomaly. No low gain antenna could provide backup capability at Pluto's distance from Earth. The current best estimate for the telecommunications equipment mass and power using components that will be qualified by 1994 exceeds the allocations set by the system engineer to maintain mass and power budgets. Members of the design team are currently evaluating the feasibility and cost to this program for development of lighter, less power-hungry components, and such development will be one focus of 1993 work.

The Electrical Power Subsystem is based on a "3-brick" RTG where each brick is derived from the standard GPHS. The RTG generates 42 W at the beginning of the mission and 38 W eight years after launch, at 10 Vdc. Power is upconverted to 28 Vdc and distributed on two busses at 10 and 28 Vdc. Capacitor banks are used on each bus to accommodate short duration spike loads such as ACS thruster pulses, switch transients, etc. Lithium primary batteries with 4 A-hr capacities are switched onto the power busses to support longer duration peak loads during propulsive maneuvers and encounter. Excess RTG power is shunted to a radiator. Power consumption of 38 W during the downlink/cruise mode includes 30% contingency and is summarized in Table 3. Losses for voltage conversion and regulation are included. Power consumption of 47 W during encounter (includes 4 W for science) and 77 W during propulsive maneuvers is accommodated using primary lithium batteries to augment the RTG. Total battery energy usage is less than 120 W-hrs. The battery may be eliminated in an updated design.

Table 3 — Spacecraft Power Summary (W).

Telecommunications	12.0
Electrical Power	7.0
Attitude Control	3.2
Command and Data Handling	5.0
Structure	0.0
Propulsion	0.5
Thermal Control	0.5
Science Instruments (replacement heaters)	1.0
Total	29.2
Contingency (30.0%)	8.8
Total Power	38.0

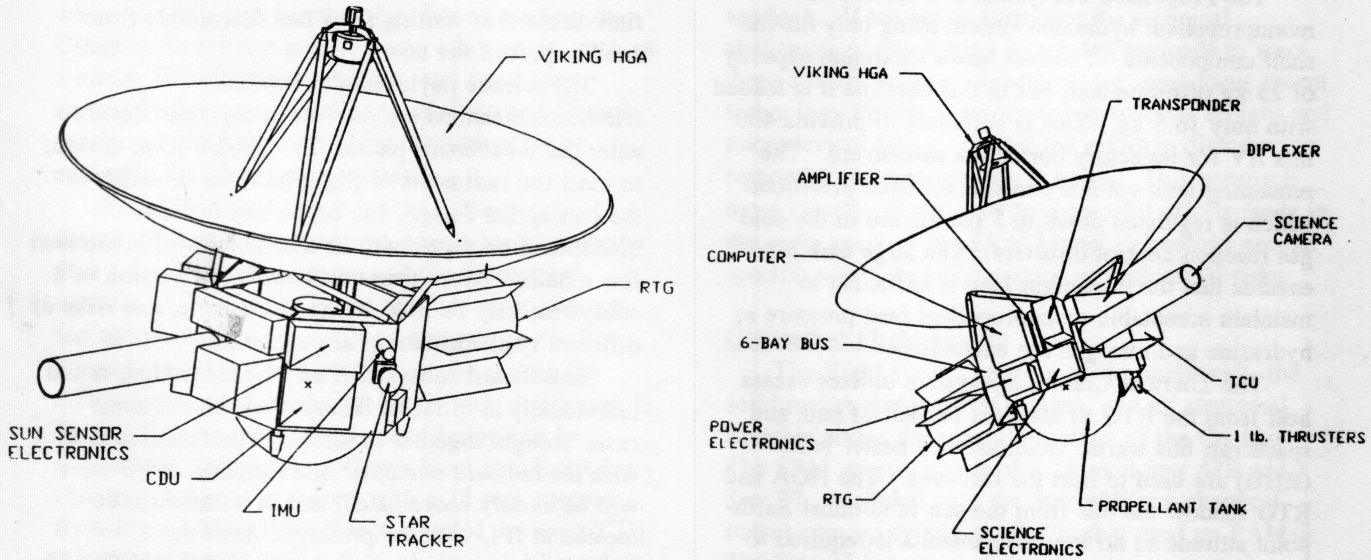


Figure 8 — PVS Spacecraft Cruise/Encounter Configuration: 8a) +Z Isometric View, 8b) -Z Isometric View.

The Attitude Control Subsystem uses a wide field of view miniature star camera for its primary inertial sensor. Star matching is done using the processor in the Command and Data Subsystem. Three solid state rate sensors are used to maintain the attitude reference of the 3-axis stabilized spacecraft between star updates. A sun sensor is included as an emergency backup reference for locating the cone angle to Earth. Control is via cold gas thruster couples about all three spacecraft axes. Pointing knowledge is 1.5 mrad, and stability is 10 μrad over one second. Fast slews of 90 degrees require 2.7 minutes, zero rate to zero rate, plus settling time.

The Command and Data Subsystem uses a 1.5 MIPS reduced instruction set computer (RISC) on a single board with radiation hardened parts (to 25 krad). VLSI (Very Large Scale Integration) ASIC (Application Specific Integrated Circuit) and surface-mount packaging technology are used for reduced mass, and power strobing is used to minimize power. Input/output is via direct lines, serial interfaces, a high rate science bus (5 Mb/s), and an engineering bus. The 200+ Mb solid state memory uses high density packaging and includes an error detection and correction capability. A data compression chip losslessly compresses the science data before storing it in memory. Further software-driven compression is being investigated for selected imaging data.

The main structure of the spacecraft is an aluminum hexagonal bus. The propellant tank is held within this structure by three brackets on its equatorial plane. Truss structures are used for the adapter to the injection stages and to mount the antenna to the bus. There are no articulated or deployed mechanisms.

The Propulsion Subsystem is a blowdown monopropellant hydrazine system using only off-the-shelf components. The tank has a maximum capacity of 23 kg of propellant, but in this baseline it is loaded with only 16.5 kg. This is sufficient to provide 400 m/s ΔV for trajectory correction maneuvers. The remaining tank volume contains nitrogen pressurant which is regulated down to 5 psi for use in the cold gas reaction control thrusters. The large tank size ensures that the pressurant load is sufficient to maintain acceptable monopropellant feed pressure as hydrazine and nitrogen are expended.

The Thermal Control Subsystem utilizes excess heat from the RTG to keep the propellant tank and spacecraft bus warm. Radioisotope heater units (RHU) are used to heat the thrusters. The HGA and RTG shadow the bus from the sun in nominal Earth-point attitude so no low-gain antenna is required to maintain a proper thermal control attitude. Multi-layer insulation (MLI), small heaters, and louvers regulate component temperatures during power fluctuations and as insulation diminishes.

Some mass reductions may be possible by trading capabilities such as data rate for amplifier power or antenna mass. Such reductions may or may not affect cost. Further reductions could be made through the aggressive use of microspacecraft technology (advanced electronics packaging, a composite propellant tank), but this would certainly increase cost. Competing with efforts to reduce mass is a desire to install fault tolerance through redundant components. A second design iteration is currently underway that will trade on the possible mass reductions and produce a fault tolerant spacecraft concept. The result will no doubt be a heavier spacecraft with higher reliability.

Assembly, Test and Integration

The PFF spacecraft is to be assembled as a series of higher fidelity hardware and software builds. This year there is a full-size, low fidelity mockup of the spacecraft, accompanied by conceptual designs for each of the subsystems and their operation. Next year, key subsystems will be created in "breadboard" form, utilizing existing, but often non-qualified hardware and software components, and even some nonfunctional components. Many early problems will be worked out at this level where components are inexpensive, different techniques may be easily tried, and reliability is not a concern. Delays introduced by problems discovered and worked out at this stage tend to be far less expensive than delays discovered later. Not only is less redesign required, but the very small early team, with fewer than ten full-time equivalent personnel, runs up costs at a slower rate to the extent that anyone is "waiting around." Cost impacts of delays later in a project development are much greater with a larger size team, members of which can often find themselves waiting for a late deliverable from another part of the team.

The science payload and downlink telecommunications are the first subsystems slated to enter the breadboard phase. Breadboard-level design, to meet the rudiments of flight-level functionality in the conceptual design, has begun and the first breadboard hardware will be assembled within the next few months. More than one breadboard version of a subsystem may be tried where the benefits and risks of different implementations are uncertain.

Breadboard subsystems may be assembled, tested individually in different laboratories, and in some cases brought together to test key interfaces, such as with the onboard computer and memory. PFF work will be closely coordinated, and in some cases co-located at JPL with the prototype Asteroid Investigation with Microspacecraft (AIM) and Caltech SURFSAT developments.

The next step is a "brassboard" spacecraft, having functional replicas of most subsystems built separately and then integrated into a partially

functional spacecraft, with supporting computers and other laboratory equipment connected. Some subsystems will not be at all functional; for example, the RTG will be replaced by a simple mechanical replica and power supplied from convertors running off wall current. Other subsystems will be closer to flight functionality and configuration, such as the computer and solid state memory, which might differ from flight versions only in lack of screened electronic parts and completeness of loaded software. In some cases, flight-qualified or -qualifiable components might take their places in the brassboard, as in the case of some structural members, the sun sensor and star cameras bought from vendors. Software will be developed and tested concurrently with hardware, using processors and other components identical to those planned for flight. Software development support workstations will become ground test workstations, and finally flight control workstations as development progresses, with their operating software being developed and upgraded through phases similar to that of the hardware.

The spacecraft brassboard will be utilized to work out nearly all the necessary subsystem interface details and problems while there is still time to modify custom flight hardware and software. This, and the key subsystem intercompatibility tests at the breadboard level, is intended to minimize more expensive redesign and rework after flight hardware is in the final production pipeline.

Two flight qualified spacecraft are presently planned, built alongside one another. The first to be completed will be subjected to system-level flight qualification testing, and refurbished for the second launch. The other spacecraft will be launched first.

Mission Operations and Tracking

Personnel at JPL and the University of Colorado (CU) at Boulder have developed a cooperative concept for low cost mission operations. JPL will provide DSN tracking and navigation and CU will develop a single, simple mission operations data system design that will have versions located in operations nodes at both Boulder and at JPL. Routine operations are proposed to be done out of Boulder with planned JPL support for critical events and as required for spacecraft anomaly analysis. Downlink is to one or more 34 m DSN stations. Only Goldstone supports Ka-band downlink at present. Because the antenna figure is not sufficiently stable, no Ka-band capability is planned for the 70 m stations, though they are expected to be used for the X-band uplink.

Operations at CU will have an educational dimension. Students, supervised by experienced professionals, will staff operational positions. Science and engineering data will be accessible to schools. Both operations nodes will be set up to allow student participation and visibility into space flight operations. Lessons learned at CU using students to operate SME (Solar Mesosphere Explorer) will be applied toward achieving low cost, efficient operations for the Pluto mission [9].

Mission and spacecraft design features are key to enabling small team operations and a relatively simple ground data system. Pluto Mission Development management is strongly committed to mission operations engineering participation in the spacecraft, instrument, and mission design process. Current design features that are important toward enabling low cost operations include:

- a spacecraft design that permits long periods of unattended operations during cruise. This enables routine cruise operations to be built around a single weekly DSN tracking and data collection pass.
- a spacecraft engineering data return strategy that exploits on-board data processing and analysis to minimize the amount of engineering data that must be downlinked and analyzed.
- spacecraft command and control capabilities that allow cruise commands to be uplinked without elaborate simulation and constraint checking.
- an encounter/flyby command sequence that is pre-planned and tested during cruise and is only "tweaked" immediately before closest approach to allow for trajectory and arrival time uncertainties.
- a large on-board memory that permits capture and storage of all the science data collected during flyby and allows its subsequent return over a limited downlink (40 b/s) via routine daily DSN passes for up to a year following encounter.

Pluto 350 Implementation Option

The Pluto 350 spacecraft design, adopted from an earlier Discovery-class proposal [10], has a net spacecraft dry mass of 316 kg with a mission cost under $400M. While larger than the PVS spacecraft, the science payload is still ambitious for this class of spacecraft. In the implementation described here, five instruments are carried, going beyond the Category "1a" measurement objectives. Instruments include a narrow-angle camera sharing fore-optics with an infrared (IR) imaging spectrometer, an ultraviolet (UV) spectrometer, a radio science subsystem with an ultrastable oscillator (USO), and an ion-neutral mass spectrometer. It is not yet clear that all these instruments can be built and accommodated within spacecraft resources and cost constraints. Characteristics of these instruments are provided below in Table 4.

Table 4 — Pluto 350 Science Payload (Characterized 5/15/92 by J. Ayon, R. Terrile and members of the NASA OPSWG.)

Instrument	Mass (kg)	Power (W)	Data Rate (kb/s)
Narrow-angle camera	12.0	5.0	tbd
with IR imaging 2 spectrometry	10.0	5.0	tbd
UV spectrometer	3.0	2.0	1.0
Radio tracking and occultation	3.6	5.0	0.4
Ion-neutral mass spectrometer	3.0	10.0	1.0
Subtotal*	31.6		
Payload allocation	30.0		
Contingency allocation (20%)	6.0		

* Payload total uses 1.6 kg of the contingency allocation.

Each instrument and its measurement objectives are described below.

Narrow Angle Camera (NAC): The PF350 imaging camera utilizes a Ritchey-Cretien design with lightweight optical components and an eight-position filter wheel. A primary aperture diameter of 12.2 cm combined with a focal length of 750 mm yields an *f*6.1 system centered about 900 nm wavelength. The focal plane incorporates a Loral 1028x1028 CCD with 7.5μm pitch. This optical configuration results in an IFOV of 10 μrad for a final FOV of ~0.6°. In addition to the visual imaging capabilities, an IR capability is being examined for incorporation. The IR spectrometer would most likely use HgCdTe detectors passively cooled to around 100K. The wavelength region of interest would be 1.0–2.5 μm with a $\lambda/\Delta\lambda$ of 200. The beam for the IR spectrometer would be picked off with a dichroic mirror prior to the visual focal plane and routed to the spectrometer. Possible options for the spectrometer design include integration of an aspheric holographic grating and a 2-D detector array with a flat focal plane. Another option under study for the NAC is the addition of a UV capability as described for the PFF instrument.

Radio Science: The radio science experiment takes place during entry and exit of Pluto's Earth occultation zone, and uses a Mars Observer USO as its principle component. To conserve mass, the radio science experiment integrates capabilities within the spacecraft transponder. Additional elements necessary for the radio science subsystem include an A/D converter, filter, and multiplier.

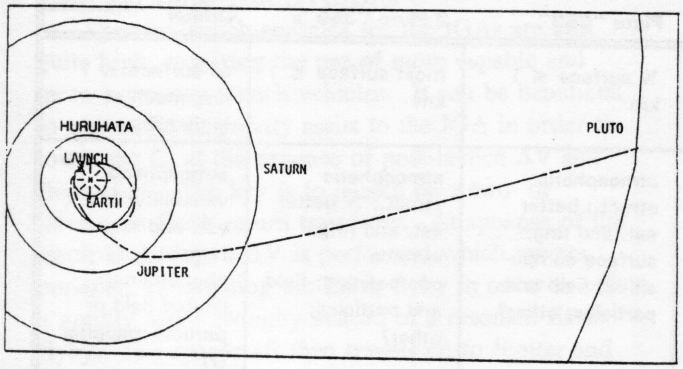
Ion and Neutral Mass Spectrometer (INMS): INMS measurements take place within 4 R_p of the planet. Navigational capability for such a close flyby is under investigation. This instrument is based on the one flown on *Pioneer* Venus Orbiter (PVO). The instrument envelope is cylindrical, approximately 15 cm in diameter by 26 cm in length. It is mounted on the bus to point into the ram (velocity) direction during the "atmospheric" transit at Pluto. The instrument will measure from 1 to 46 amu with a discrimination of ≤ 1 amu. Measured species include hydrogen, methane, nitrogen, carbon monoxide, and argon. Measurement would be sequenced between ion and neutral modes. Improvements over the PVO design include lightweight materials and custom VLSI electronics to reduce mass and power requirements.

Ultraviolet Spectrometer (UVS): Several options for the spectrometer design are being considered. To achieve the goals of this solar occultation experiment, one option is to mount the UVS boresighted with the HGA so the UVS aperture would look through a hole in the antenna. This instrument would have a simple fore-optic design (pinhole) allowing it to stay within mass allocations. As mentioned earlier, options for adding UV capability to the NAC are also being considered. Observations in the 1000Å to 2000Å regime with a $\lambda/\Delta\lambda$ of 1000 are the goal.

The Pluto 350 implementation uses an Earth and subsequent Jupiter gravity-assist trajectory, as shown in Figure 9. Encounter V_∞ is 12.8 km/s. The 13.6 year flight time requires strict attention to fault tolerant design, and therefore a "Class-A" dual-string functional reliability philosophy has been adopted.

Like the PVS spacecraft, the Pluto 350 design uses the *Viking* orbiter 1.47-m HGA. The current spacecraft concept is shown in Figure 10. The 316 kg net dry mass of the spacecraft includes 60 kg of contingency. There is 111 kg of hydrazine monopropellant. Spacecraft structure is based on a five-bay bus (hexagon with one side removed) with monocoque tank support and launch vehicle adapter interface. Thermal control utilizes 15 layer thermal

blankets on the spacecraft flat surfaces and 10 layers on the RTG support structure. Bus radius is 0.67 m and height 1.9 m.



Launch	2001/11/15
Asteroid Huruhata Flyby (V_∞ = 10.3 km/s)	2002/12/04
Propulsive Maneuver (ΔV = 540 m/s)	2003/05/23
Earth Flyby	2005/01/01
Jupiter Flyby (15.5 R_J)	2006/05/10
Pluto/Charon Encounter	2015/06/27

Figure 9 — Pluto 350 Heliocentric 3+ΔVEJGA Trajectory.

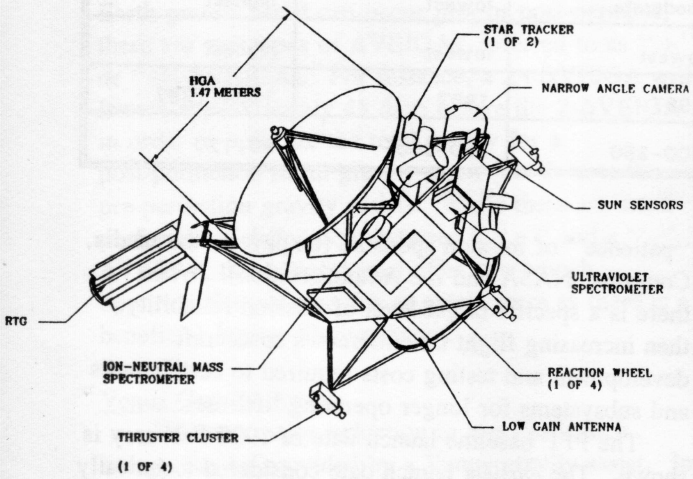


Figure 10 — Pluto 350 Spacecraft: Cruise and Encounter Configuration.

Instrument and telecommunications pointing are achieved by pointing the 3-axis stabilized spacecraft at the target of interest; instruments are body-fixed with no articulated platforms. Thruster clusters are also body-mounted. The attitude control system utilizes two off-the-shelf star trackers (one for redundancy) and the *Magellan* inertial reference unit (SKIRU III).

Data rates at Pluto range from 193 to 516 b/s with X-band downlink to a 70 m DSN station. Near encounter data is stored for later return. The Command and Data system is based on a 2 Gb solid state memory with 3-D packaging of 4 Mb RAM chips for data storage. Power is supplied by a modular RTG supplying 248 W at launch and 169 W 15 years later.

Option Comparison and Tradeoffs

A comparison of different Pluto exploration options is shown in Table 5. Options compared are the Pluto Fast Flyby using either one or two launches, the Pluto-350 with one or two launches, and the Pluto Orbiter using a single launch. Comparisons are made for the baseline options as of 1992 April 27. In all probability, the selected baseline will utilize a spacecraft somewhere between the PVS and PF350 in size, with a greater science yield than the PFF option described here, in order to cover all Category "1a" objectives. Because of the long flight time, the Pluto Orbiter is not expected to receive further consideration without a plausible low-cost scheme to shorten flight time.

Cost

Some clear options are emerging for how to perform the first reconnaissance of Pluto, Charon, and their environs. The first option, considered feasible until little more than a year ago, was a large, Cassini-class, many-instrument flyby arriving in 2015 or later. It conceptually carried a "small" sub-probe whose arrival would be phased with the parent to image the side of Pluto not illuminated during the parent flyby. In the present fiscal climate, likely to prevail for many years, cost has replaced performance at the top of the list of guiding principles in project development. Because of accounting changes at NASA and the Department of Energy, total end-to-end mission cost is becoming more important than with prior programs, where typically cost of spacecraft and instrument development and procurement, through 30 days after launch, was the primary measure. Now "cost" attributable to a particular mission includes spacecraft development and procurement with mission operations system development costs as before, plus launch, RTG and mission operations costs.

Time

For any major mission expenditure, value received is still interpreted to a large extent based on the importance of the scientific questions addressed, and the quality and quantity of data yielded which enables addressing these questions. Time is a major component of value, and of cost as well. Answers to key scientific questions sooner are more valuable than

Table 5 — Pluto Exploration Baseline Option Comparison.

Attribute	Flight Option				
	Fast Flyby	2 Fast Flybys	Pluto "350"	2 Pluto "350"s	Orbiter
Science Yield	½ surface ≤ 1 km	most surface ≤ 1 km	½ surface ≤ 1 km	most surface ≤ 1 km	all surface ≤ 1 km, much at higher res.
	atmospheric struct.; cursory sat. and ring	atmospheric struct.; cursory sat. and ring	atmospheric struct.; better sat. and ring; surface composition; field and particles; other?	atmospheric struct.; > better sat. and ring; surface composition; field and particles; other?	atmospheric variability; full sat. and ring; temporal phenomena; limited field or particle mapping (?)
Cost: spacecraft launch veh. mission ops RTG	lowest highest lowest lowest	~ × 1.2 ~ × 1.75 (?) ~ × 1.25 (?) ~ × 1.5 (?)	highest lowest moderate highest	~ × 1.2 ~ × 1.75 (?) ~ × 1.25 (?) ~ × 1.5 (?)	moderate highest highest (?) low–moderate
Launch Launch flexibility	1998 February any year	1999 February any following year	2001 November 2001 ±1 (difficult)	2002 January ~45 d > L#1	2001 November 2001 ±1 (difficult)
Data return	≤ 2005	≤ 2006	2013–2016	2013–2016	2019–2021
Flight time (yrs) launch to enc.	7	7	13.5	13.5	18–20
Mission risk	high	low–moderate	moderate	lowest	highest
Technical risk	moderate	moderate	lowest	lowest	highest
Technology date	1994	1994	1997	1997	1994–1997
Dry mass (kg)	75–100	75–100	300–350	300–350	90–120

later, and to some extent data quantity and quality may be tradeable for time. Such trades are very difficult to quantify. Pluto's pending atmospheric collapse puts its own imprint on the timing question.

The longer we spend preparing for a mission, the greater the cost will be, as people's efforts get devoted to "what-if" studies, "improvements," tradeoff analyses and the like. Given time, we can always make the product better, at a cost. Our question is what is "good enough," and then how do we proceed rapidly with implementation with a minimum of deviations. Cancellation of programs such as the A-12 and P-7 aircraft, and some space projects, has shown that high-capability programs can be endangered for a variety of reasons. Sometimes insisting on the most performance gets a program nowhere beyond paper. On the other hand, simply sending a tracking beacon by Pluto will interest no one, so we must reach a balance of value, cost, and speed of implementation.

Flight time is another important criterion for comparison, as it directly impacts mission operations cost, probability of mission success, and the "patience" of mission sponsors (taxpayers, the media, Congress, NASA and the Administration). When there is a specific target level of mission reliability, then increasing flight time increases spacecraft development and testing costs required to certify parts and subsystems for longer operating lifetimes.

The PFF baseline launch date of 1998 February is shown. The earliest launch date considered technically attainable is 1997, with the primary limitations being lead time to develop and procure the RTG, and availability of funding based on other priorities. With launch periods every year, a second spacecraft can follow the first by approximately one year, with arrival "fine-tuned" by some odd multiple of half Pluto's presumed rotational period after the first encounter, to reveal the sides of Pluto and Charon in shadow during the first encounter.

PF350 is best launched 2001 November. A major problem could postpone launch about one year, which would necessitate different propellant loading and trajectory design. In the case of minor problems that could be corrected within about six weeks, the

spacecraft could be launched during the "3-minus" ΔVEJGA opportunity. Alternatively, the "3-minus" opportunity could be utilized to launch a second spacecraft, with arrival at Pluto phased as noted above.

Selection of a mission "technology date" is made by indicating a time before which certain components or techniques must be certified or demonstrated for space flight. This deliberately limits or opens up the designers' choices of materials, parts, subsystems and vendor equipment. Earlier dates generally mean less performance is expected to be available for a given mass, power or cost. Later dates introduce risks that promised performance will be unavailable from ongoing technology development efforts, with costly and time-consuming redesign a possible consequence. Project managers have some ability to accelerate development or certification of key items through either project funds or persuasive cooperation with technology development efforts, which have other sources of funds. By the same token, technology development managers look for advocacy and endorsement from high visibility projects to elevate the priority of their own efforts. Communication between these two groups is increasing, as it can be beneficial to all involved if the risks are understood and accounted for.

Scientific Yield

Once the Category "1a" measurement objectives are covered, the metrics for comparison of different mission implementation options will be: 1) to what extent will the depth of the data minimize remaining ambiguities of surface geology and composition, and atmospheric composition, 2) what lower-priority objectives will be covered as a consequence of the measurements made to satisfy the higher priorities, or with minimal operational consequences, 3) what geographic and temporal coverage will be provided as a consequence of flyby speed, geometry, and the number of encounters (one spacecraft vs. two), and 4) how can the value of early, less comprehensive data be compared with the promise of more data as much as a decade or more later?

Risk

Mission risk is a qualitative characterization of the probability that a given mission (which may comprise one or more spacecraft) will not satisfy the primary mission objectives. This risk can be divided into categories of launch risk, governed by the probability of failure of the launch vehicle and its upper stages, and spacecraft risk, governed by the probability of a crippling spacecraft failure before some or all of the Pluto encounter data is transmitted to Earth.

Launching two spacecraft separately is the simplest, though not necessarily the least costly, way to reduce mission risk. The Titan family of launch vehicles has a historical success rate of around 90%, while the Delta and Atlas operate closer to 95%. Neither number considers the risk of failure of any stages atop the standard launch vehicle, nor the spacecraft propulsion system.

Qualitative comparisons have been made based on dual spacecraft carried on a single launch vehicle, but the flight time penalty is considered too great. Such an arrangement is possible for either flyby mission. Typically, a two-spacecraft-in-one-vehicle launch, while subjecting both spacecraft to launch risk, also increases post-launch risk because the spacecraft take longer to reach their destination, allowing more time for failures to occur.

Technical risk is a qualitative characterization of the probability that the technologies required to successfully build and operate the spacecraft described will be unavailable by the technology date considered as the cutoff for incorporating flight certified components and subsystems into the final spacecraft design. There is some softness in this risk, especially in that delays in flight certification may be acceptable if sufficient progress is shown toward that goal. Work-arounds may also involve acceptance of degraded performance (e.g., utilize a smaller solid state memory within the mass allocation), or by allowing mass growth of the equipment required to perform a particular function. Such mass growth would utilize contingency and lengthen flight time; sufficient contingency may be unavailable by the time the final design needs to be frozen.

Programmatic Issues

Reliability and Product Assurance

An analysis of the *Viking* Orbiter, *Voyager*, *Magellan* and *Galileo* missions examined their flight histories for failures which, had they occurred on a spacecraft without redundant subsystems, would have terminated the mission [11]. To date there have been seven such failures (see Tables 6a and b, from the reference), all of which appear to be random, in that no single mission-terminating type of failure has been repeated in another redundant "string" or on the other spacecraft. If this were to hold true for spacecraft sent to Pluto, then doubling the number of spacecraft (from one to two) could be expected to double the probability of at least one spacecraft successfully completing the mission.

Analysis shows that such failures have occurred at a rate which decreases with time. In contrast, serious failures occur at a higher rate on the ground during test and checkout before launch. Once a spacecraft makes it through the rigors of ground handling and launch, then the probability of failure during any fixed

Table 6a — Planetary Mission Inflight Failure History: Operating Hours and Number of Failures.*

Mission	Launch Date	Number of Spacecraft	Number of Relevant Spacecraft	Flight Hours	Relevant Flight Hours	Relevant Failures*
Viking	1975	2	4	18,000	72,000	0
Voyager	8/77	2	4	125,000	500,000	3
Magellan	4/89	1	2	26,000	52,000	4
Galileo	10/89	1	2	22,000	44,000	0
Total	1975-1989	6	12	191,000	668,000	7

* One year = 8,766 hours
Relevant failure = failure which would have resulted in loss of mission on a single-string spacecraft.

length of time decreases as the mission proceeds at least for the mission durations observed to date.

Strict practices have assured a high quality of parts and workmanship on planetary missions over the last two decades [12]. However, these same strict practices have also bred a culture of costly design, parts selection, and testing. Work is in progress to assess which practices actually add value to a spacecraft in terms of its reliability, and which may do more to add "management comfort" without much actual reliability improvement. In some cases, system level testing of the final product may be able to take the place of one or more lower levels of testing, at the risk of high rework costs in the event failures are found [13]. Also, it may be possible to better allocate reliability and product assurance resources (dollars, staff talent, test time and spacecraft mass) where the risks are the greatest. JPL procedures are presently under review with the expectation that they will be substantially streamlined.

Reliability is important to any space mission, but the long durations of Pluto missions makes them especially sensitive. Small spacecraft, like the PFF concept, involve very direct tradeoffs between mission risk (which can be lowered by adding redundancy at the expense of mass of additional components) and expected scientific yield (which can be increased by adding instrument mass and capability, which comes in place of redundancy if mass is fixed). At 63 kg, *Pioneers 6, 7* and *8* are still operating. *Pioneer 6* was launched in 1965—27 years ago!

With the limited number of missions in any reliability database, the constantly changing selection of components and design practices over time, and the design differences between any Pluto spacecraft and its predecessors, there appears to be no way to accurately predict quantitative mission reliability. At the same time, depending on one's assumptions, there are many seemingly credible ways to massage what little data exists and come up with any sort of quantitative reliability number one might want. No such number will be a rigorous reflection of reality. It will be possible to compare relative reliability of slightly different design options, but not to assure achievement of some absolute reliability level. Therefore, mission risk will have to be mitigated within available resources using some combination of "best practices" and "engineering judgement," at the end of which the project manager will have to lay his or her reputation on the line and give approval to launch.

Table 6b — Failure Analysis for Relevant Failures Noted in Table 6a.

Mission	Failure	Hours into Flight*
1 Magellan	Gyro	3,624
2 Voyager	Receiver	5,496
3 Magellan	Comp. memory	11,112
4 Magellan	Data storage	12,480
5 Magellan	Transmitter	20,000
6 Voyager	FDS memory	35,800
7 Voyager	Transmiter	88,400

* One year = 8,766 hours.

Radioisotope Safety

Pluto's vast distance from the sun renders solar power impractical, making the RTG seem at this time the logical power source. Because missions baselining the use of RTGs may have the potential, in the event of a severe launch accident, to impact the well-being

of some portion of the environment and its inhabitants, such missions must satisfy two independent regulatory processes geared toward maximizing safety: the National Environmental Policy Act (NEPA) compliance process and the launch approval process.

NEPA requires that an environmental assessment or environmental impact statement (EIS) be prepared for any Federal project with the potential to significantly impact the environment. NASA's guidelines for implementing NEPA regulations require that an EIS be prepared for the "development and operation of nuclear systems, including reactors and thermal devices used for propulsion and/or power generation" [14]. The EIS provides the public and the cognizant NASA decision maker with information on the purpose and need for the proposed action, reasonable alternatives to the proposed action, and the environmental impacts associated with the proposed action and the reasonable alternatives to it. To adequately discuss potential environmental impacts, however, a baseline design has to be developed to a degree that allows both a meaningful characterization of potential environmental hazards and a comparison with reasonable alternatives. The analyses and implementation options discussed earlier in this paper constitute the first step in developing this baseline design. The baseline to be considered in the EIS is not to be confused with the preliminary, multiple "baselines" described elsewhere in this paper. More design maturity is required for the EIS. Once this baseline design has been developed and studies have been conducted to identify and assess reasonable alternatives and potential environmental impacts, NASA will produce and publish a draft EIS for public comment. Based on this comment, NASA will then issue a revised EIS that will serve as the basis upon which the cognizant NASA decision maker will decide whether to proceed with the proposed action or select one of the alternatives to the proposed action. The decision will be formalized in a public Record of Decision (ROD).

In addition to the NEPA compliance process, Presidential Directive PD/NSC-25 established the launch approval process in 1977. According to this directive, the Executive Office of the President must approve launches of spacecraft utilizing nuclear power sources. To obtain this approval, the launching agency must first specify spacecraft and launch accident hazard environments that might jeopardize the nuclear power source's integrity. Based on the specification of these environments and the results of its RTG safety testing program, the Department of Energy (DOE) must then perform a series of safety analyses. The NASA Administrator will use these safety analyses as the basis for deciding whether or not to request launch approval—a request which, if made, is then submitted to the director of the Office of Science and Technology Policy (OSTP). In the meantime, NASA must confer with the Department of Defense (DOD) and DOE to form an Interagency Nuclear Safety Review Panel (INSRP). This panel independently reviews DOE's safety analyses as they are developed and reports its findings to OSTP in the form of a Safety Evaluation Report (SER). After receiving a request for launch approval, OSTP reviews the findings of the SER and DOE's safety analyses and makes a recommendation on whether or not to grant launch approval. Either the director of OSTP or, if he or she feels it advisable, the President makes the final decision on whether or not to grant launch approval.

The Pluto Team is currently exploring ways to maximize safety and minimize cost by applying past launch approval analyses to the early detection and correction of spacecraft and launch vehicle design issues pertinent to RTG safety. In so doing, the team hopes to develop a cost-effective spacecraft and launch vehicle baseline well within the safety bounds associated with past missions.

Accelerated Procurement

Federal procurement laws and regulations are often cited as a serious impediment to rapid mission implementation. Pluto mission development cannot succeed within prescribed cost constraints in a business-as-usual mode for procurement, or for many of the other processes required to bring a mission to successful fruition. From the preparation of the first proposal to NASA Headquarters, the Pluto Team has included a procurement specialist working together with trajectory, hardware, mission operations designers and other project personnel to help resolve procurement-related problems before they become serious enough to cause delays. The Pluto Team has begun to incorporate lessons of some of the SDIO-sponsored missions and the Miniature Seeker Technology Integration (MSTI) mission into its planning. Rapid response can be achieved totally within the letter and spirit of the law. Given the proper motivation of personnel, delays brought about by internal procedures can be minimized.

Summary and Continuing Direction

Pluto's distance makes any mission there a challenge. Once considered incredibly remote, Pluto is now clearly within reach. As the last known unexplored planet, arguments for a visit have become more compelling since *Voyager's* success. What measurements we have been able to make from Earth render Pluto and Charon an enigmatic pair, and there is no doubt that important pieces to the puzzle of our own Solar System's formation will be revealed with a successful initial reconnaissance.

The key questions of implementation are: 1) what kind of science-driven measurements make sense, 2) how attractive can we make the price of the mission, and 3) how long should we wait to learn what Pluto chooses to reveal to us? Other key questions involve acceptable levels of reliability, utilization of newer technologies for smaller spacecraft than we have grown accustomed to using for recent planetary missions, and how best to make do with a modest total data return limited by distance and onboard memory capacity.

Work in progress is aimed at selecting the most rapid and lowest cost possible mission execution which will be considered scientifically worthwhile. The Pluto Very Small and Pluto 350 options represent perhaps two ends of a very cost-constrained spectrum, with considerable generic differences. While total costs are likely to be similar, there are high-visibility tradeoffs between spacecraft and launch vehicle costs, science capability, and which decade we get our data back in. There is also the significant danger that if implementation is delayed, costs will rise, interest will wane, and there will be less to learn about a diminishing Plutonian atmosphere.

Drawing from the work to date, the baseline in preparation is driven by the goal of soonest possible arrival at Pluto, with the following conditions: 1) new start line item in NASA's budget for FY1996, 2) investigation of launch aboard Proton, and 3) satisfaction of the Category 1a science objectives, including imaging nearly all of Pluto and Charon at 1 km/pixel or better resolution.

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References

1. S.A. Stern, "The Pluto-Charon system," in G. Burbidge, D. Layzer and J.G. Phillips, eds., *The Annual Review of Astronomy and Astrophysics (1992),* Annual Reviews, Inc., Palo Alto, CA, 1992, in press.

2. E.F. Young and R.P. Binzell, "Comparative Mapping of Pluto's Sub-Charon Hemisphere: Three Least-squares Models Based on Mutual Event Light Curves," AGU/Montreal, and *Icarus,* 1992, in preparation.

3. S.S. Weinstein, "Pluto Flyby Mission Design Concepts for Very Small and Moderate Spacecraft," AIAA paper 92-4372, to be presented at the AIAA/AAS Astrodynamics Meeting, Hilton Head, S.C., August 10–12 1992.

4. Todd J. Bayer, et al., *Expendable Launch Vehicles Summary for JPL Mission Planning,* JPL Document D-6936, Rev. A (internal document), February 1991.

5. Ross M. Jones and Christopher G. Salvo, "Microspacecraft Technology for Planetary Science Missions," Congress of the International Astronautical Federation paper IAF-91-051, 1991.

6. Ross M. Jones, "Small Spacecraft Activities at JPL," Utah State University Conference on Small Satellites, Logan, Utah, August 26, 1991.

7. Robert L. Staehle, "Small Planetary Missions for the Space Shuttle," Amer. Astronautical Soc. Annual Mtg., Paper 79-288, Los Angeles, 1979 October 29.

8. John C. Beckman/JPL, private communication to Robert Staehle and Stacy Weinstein, 1991 November 15.

9. Elaine Hansen, "Lowering the Costs of Satellite Operations: Lessons Learned from the Solar Mesosphere Explorer (SME) Mission," Paper AIAA-88-0549, AIAA 26th Aerospace Sciences Meeting, Reno, Nevada, January 11–14, 1988.

10. David H. Collins, "Pluto Flyby Study," Presentation to the Discovery Program Science Working Group, (internal document), Washington, DC, 16 May 1990.

11. Ralph F. Miles, Jr., "Pluto Mission Reliability Report: Phase I," JPL IOM 311.2-441 (internal document), April 10, 1992.

12. Anonymous, Flight Equipment Classifications and Product Assurance Requirements," JPL document D-1489, Rev. B, (internal document) January 1990.

13. Daniel S. Goldin, *Address to the 11th Aerospace Testing Seminar,* prepared text, October 10, 1988.

14. United States Government, *Code of Federal Regulations.* Title 14, Aeronautics and Space. Subpart 1216.3—Procedures for Implementing the National Environmental Policy Act. Washington, D.C. U.S. Government Printing Office, 1990.

Pluto Mission Fact Sheet

Current as of 1992 August — Subject to Change

NASA's Solar System Exploration Division is sponsoring an investigation of a possible mission to finish humanity's initial reconnaissance of our Solar System. Two lightweight spacecraft could be launched before the end of the decade on fast trajectories to reach the Pluto/Charon system in 7–10 years. This would allow characterization of Pluto's tenuous atmosphere before it collapses into frozen frost as Pluto moves farther from the Sun.

Science Objectives

- *Top priority:* characterize global geology and geomorphology; map surface composition; characterize the neutral atmosphere
- *Other objectives:* surface and atmosphere time variability; stereo imaging; high resolution terminator mapping; selected high resolution surface composition mapping; characterization of ionosphere; search for neutral gas species; search for Charon's atmosphere; determination of bolometric bond albedos; refinement of bulk parameters (radii, masses, densities); additional satellite and ring search

Candidate Science Investigations

- Visible light camera
- Infrared mapping spectrometer
- Solar occultation ultraviolet spectrometer
- Radio science uplink occultation (surface temperature and pressure profile)

Preliminary Spacecraft Specifications

- Mass: 164 kg with propellant
- Power: 61 W, supplied by a radioisotope thermoelectric generator (RTG)
- Data rate: 25–40 bps at encounter, X-band
- Launch vehicle candidates: Titan IV/Centaur + upper stages; Russian Proton + upper stages

Participating Organizations (with NASA sponsorship)

- Jet Propulsion Laboratory, California Institute of Technology
- University of Colorado
- Southwest Research Institute

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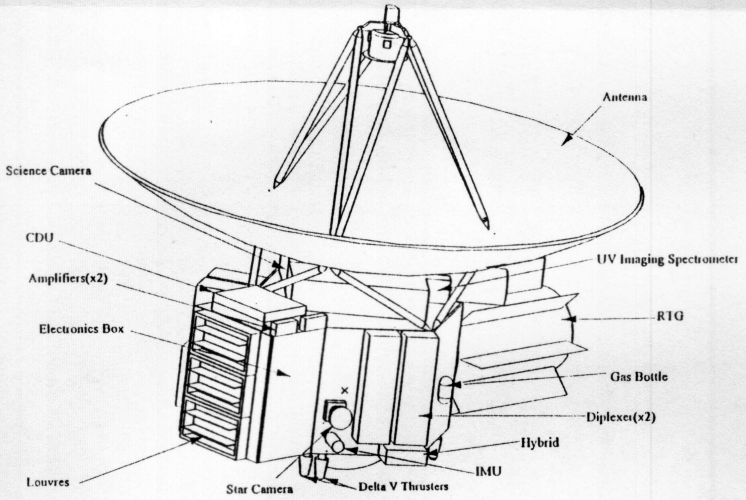
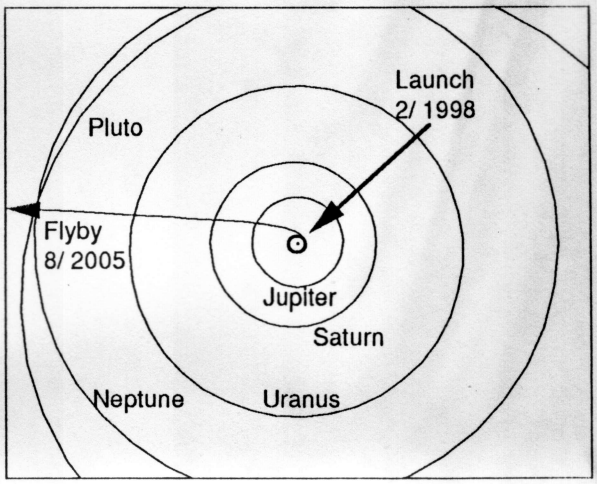
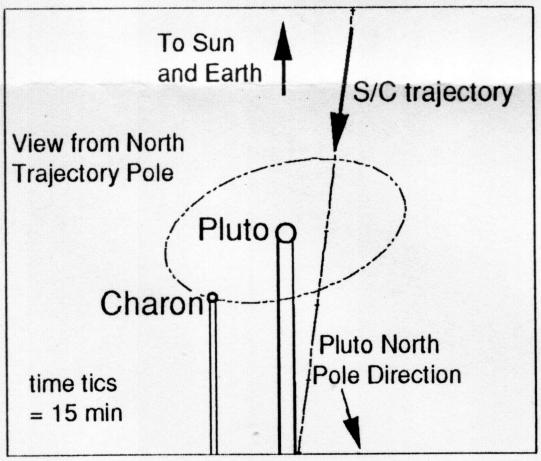


Spacecraft Cruise/Encounter Configuration. (Antenna diameter = 1.47 m = 4.8 ft.)



Spacecraft Interplanetary Trajectory. (Pluto orbits the Sun once every 248 years.)



Pluto Flyby Trajectory. (Charon orbits Pluto at ~20,000 km radius, once every 6.4 days.)

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